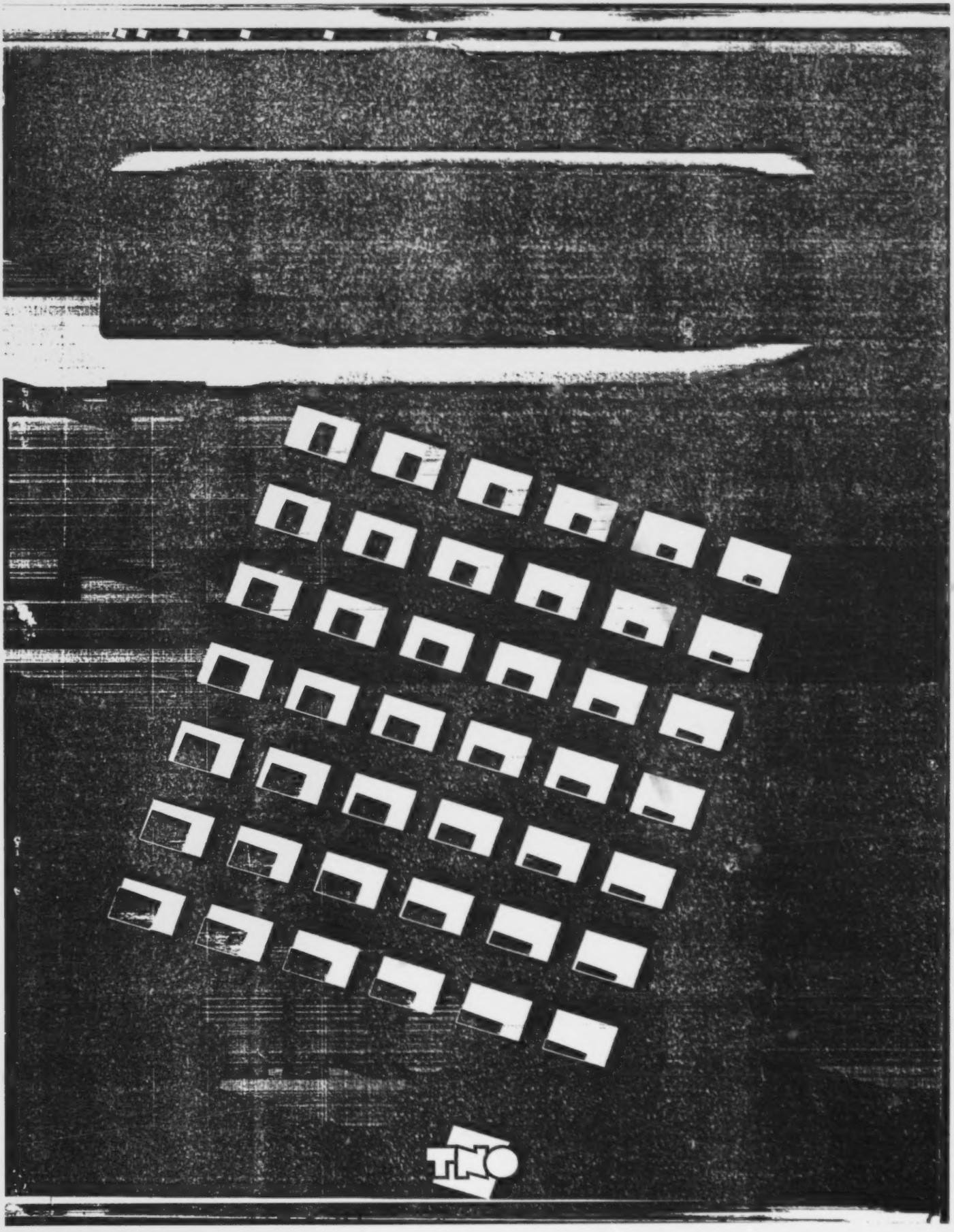
TNO

TNO report
PML 1998-A85

Integral Rocket Ramjet Demonstration Progamme (IRRDP). Executive summary report

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IRRDP 1994 - 1997

INTEGRAL ROCKET RAMJET DEMONSTRATION PROGRAM

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Résumé

Entre 1994 et 1997, le Centre de Recherches pour la Défense Valcartier (CRDV), au Canada, et le TNO Prins Maurits Laboratory (TNO-PML), aux Pays-Bas, ont travaillé en collaboration sur une activité appelée "Integral Rocket Ramjet Demonstration Program" (IRRDP) afin d'améliorer leurs bases de technologie en propulsion de missiles. Ce rapport donne un sommaire des réalisations de ce programme.

Les technologies qui ont été étudiées dans le cadre de l'IRRDP sont critiques pour la propulsion d'un missile par statofusée. En remplaçant un moteur-fusée par un statofusée, la performance cinématique d'un missile s'améliore beaucoup. Ceci cause une augmentation de la zone de létalité du missile, une portée plus importante, et la possibilité de manoeuvres à haute vitesse pour une période de temps prolongée. Pour cette raison, la propulsion des missiles par statofusée devient de plus en plus populaire.

L'IRRDP s'est concentré sur l'application d'un statofusée pour un missile de moyenne portée lancé à partir des airs. Pour ce faire, les exigences pour les missiles de moyenne portée ont été établies et des logiciels développés pour faire la conception et calculer la performance des missiles propulsés par statofusée. L'écoulement dans les entrées d'air a été modélisé, et des techniques étudiées pour isoler le foyer de combustion de l'entrée d'air pendant la phase d'accélération par le moteur-fusée d'appoint. Un propergol solide pour le moteur-fusée d'appoint et un carburant solide pour le générateur à gaz ont été développés, fabriqués, et testés dans des moteurs expérimentaux. L'écoulement dans le foyer de combustion a été modélisé par ordinateur et étudié avec un tunnel à eau. Les résultats ont fourni beaucoup de données à utiliser pour la modélisation de performance pour n'importe quel missile propulsé par un moteur-fusée ou un statofusée. Ceci comprend l'analyse des menaces et l'évaluation des missiles futurs pour les Forces Armées des Pays-Bas et du Canada.

Dû au succès de ce programme, un programme suivant d'une durée de trois ans a commencé et qui se concentre sur l'amélioration du modèle de performance existant.

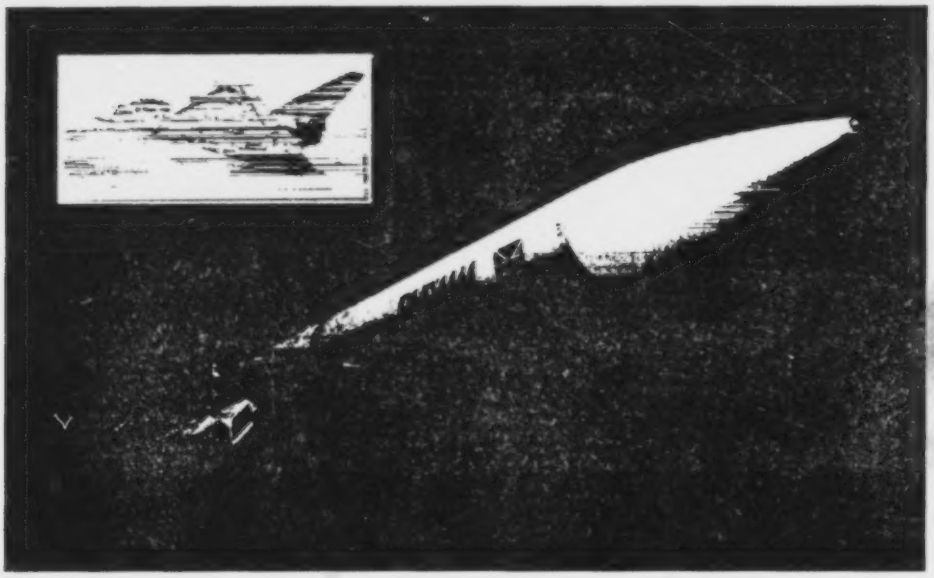

Figure M1: Pour satisfaire les besoins du nouveau missile air-air du Royaume-Uni (UK Staff Requirement 1239), incluant une performance cinétique améliorée, le BVRAAM (Beyond Visual Range Air-to-Air Missile) exige un statoréacteur ou un statofusée comme système de propulsion.

Figure M1: Ramjet propulsion is used more and more in new missile systems to enhance the missile's kinematic performance (i.e. larger No-Escape Zone, better stand-off capabilities and higher manoeuvrability. An example is the BVRAAM (Beyond Visual Range Air-to-Air Missile) that will be equipped with a ramjet to be able to fulfil the very challenging UK Staff Requirement 1239.

Managementuittreksel

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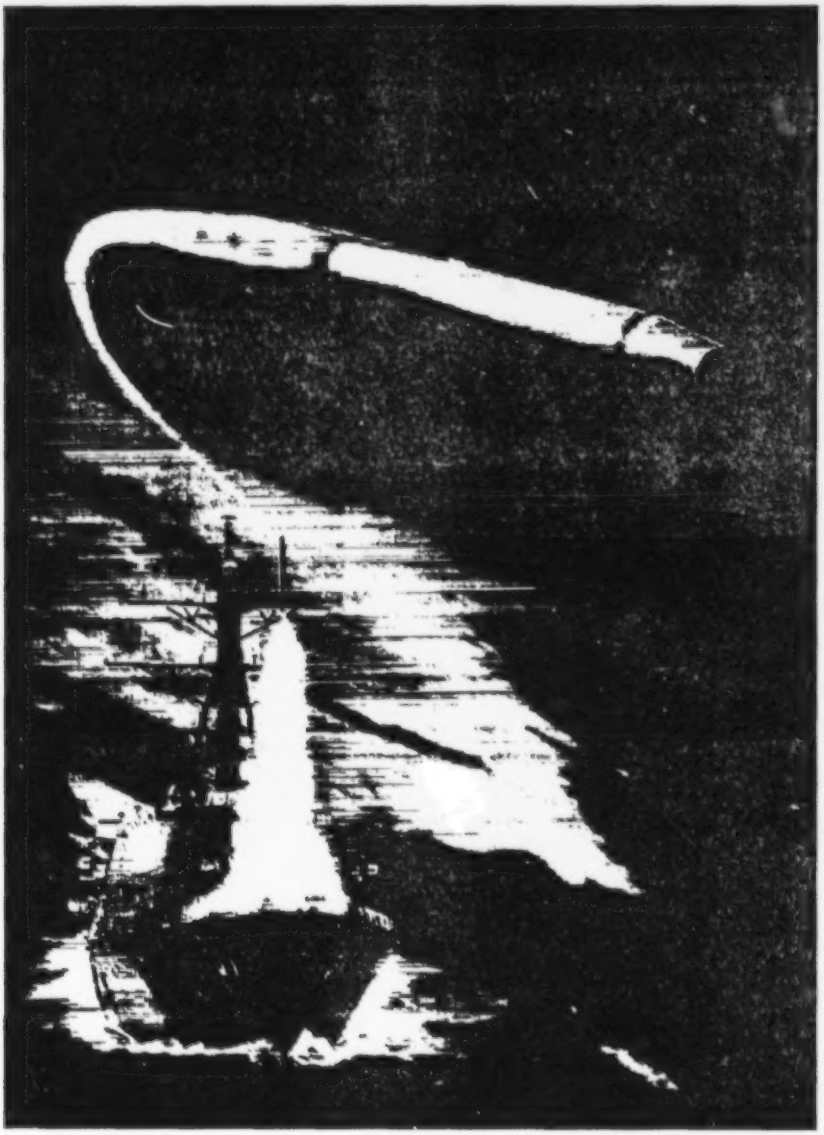
Tussen 1994 en 1997 heeft TNO Prins Maurits Laboratorium (TNO-PML) in samenwerking met het Canadese 'Defence Research Establishment Valcartier' (DREV) een gezamenlijk onderzoeksprogramma uitgevoerd met als doel haar technologiebasis uit te bouwen op het gebied van missilevoortstuwing. Dit programma staat bekend als het 'Integral Rocket Ramjet Demonstration Programme', kortweg IRRDP. Dit rapport geeft een overzicht van de uitgevoerde activiteiten.

In het IRRDP zijn de technologieën onderzocht die kritisch zijn voor de voortstuwing van missiles door ramjets. Ramjets bieden de mogelijkheid om de kinematische prestaties van een missile te vergroten. Dit leidt tot een grotere 'No-Escape Zone', een grotere 'Stand-off' range en de mogelijkheid om gedurende langere tijd bij hoge snelheden te manoeuvreren. Het is dan ook niet verwonderlijk dat ramjetvoortstuwing steeds vaker toepassing vindt in moderne missiles (zie figuur M2).

Het IRRDP heeft zich tussen 1994 en 1997 gericht op het conceptuele ontwerp van een Medium Range Air-to-Air Missile met ramjetvoortstuwing. Hierbij zijn onder meer booster- en ramjetbrandstoffen ontwikkeld, gefabriceerd en vervolgens getest in een experimentele motor. Deze resultaten hebben geleid tot een experimentele basis voor een missile-prestatiemodel dat kan worden ingezet voor de prestatie-evaluatie van raket- of ramjetaangedreven missiles. Hierbij moet worden gedacht aan dreigingen en nieuw aan te schaffen missiles door de krijgsmacht.

Het succes van het programma heeft inmiddels geleid tot de start van een driejarig vervolgprogramma dat er op is gericht het bestaande missile-prestatiemodel te verbeteren.



Figuur M.2: Door de hoge gemiddelde vluchtsnelheden van ramjet-aangedreven missiles, kunnen met dergelijke systemen op grote afstand 'time-critical targets' worden uitgeschakeld. Een voorbeeld hiervan is het Fasthawk landstrike missile voor het uitschakelen van kustdoelen vanaf zee. Fasthawk is op dit moment in ontwikkeling (range 1200 km, warhead 300 kg).

Figure M.2: Due to the high average flight speeds, a ramjet powered missile is well suited to attack and time-critical targets over a long distance. An example is the sea-based Fasthawk landstrike missile for the elimination of coastal targets. Fasthawk is presently under development (range 1200 km, warhead 300 kg).

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1 Introduction

The Defence Research Establishment Valcartier (DREV) in Canada and the TNO Prins Maurits Laboratory (TNO-PML) in the Netherlands have undertaken a collaborative activity known as the Integral Rocket Ramjet Demonstration Program (IRRDP) to investigate the key technologies of airbreathing missile propulsion through the design, fabrication, and testing of a solid fuel ducted rocket engine with an integral booster for a medium-range air-launched missile. Falling under the CA/NL Memorandum of Understanding on Co-operation in Defence Science and Technology, the resources expended on each side were approximately equal, being equivalent to one million NLG or $750K Canadian. The program's goal was to develop a technology base for the Canadian and Dutch Armed Forces in order to meet the following objectives [1]:

- to improve the operational capabilities of the Armed Forces, in terms of threat assessment, foreign weapons evaluation and potential applications;
- to be able to assist the Armed Forces in decisions concerning future purchases of weapon systems equipped with airbreathing propulsion;
- to be able, if an international industrial effort is initiated, to participate in future multi-national (airbreathing) propulsion projects for missile development programs on a basis of (industrial) expertise and capability and demonstrated international co-operation.

The following pages describe each of the main accomplishments of the IRRDP in more detail, including descriptions of how the work was carried out and examples of the results.

2 Requirements

The majority of modern missiles are still propelled by solid propellant rocket motors. For applications requiring long range and/or high average velocity, airbreathing motors offer performance gains that cannot be met by solid propellant rockets. A key objective of the IRRDP was to evaluate the performance gain of a reference missile concept when its solid rocket motor propulsion system is replaced by an integrated rocket ramjet engine. A schematic system breakdown of a ducted rocket is shown in Figure 1. The subsystems of the rocket ramjet-propelled missile are the gas generator, the ramjet combustor, the integrated booster, the port covers, the intake and the mass flow controller (valve).

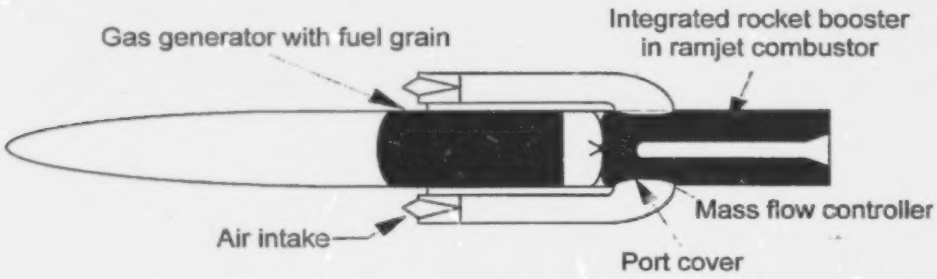
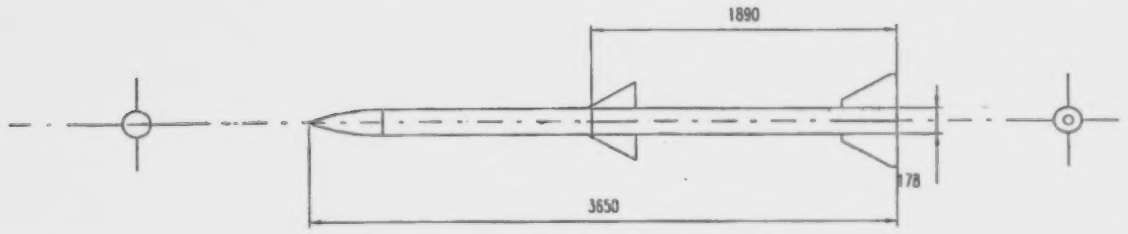


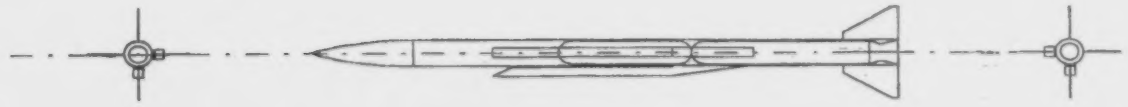
Figure 1: Schematic of an integral rocket ramjet.

Two reference air-to-air missiles were described in detail [2] as a start for the performance simulations with the propulsion performance model. Originally, each reference missile used a solid rocket motor with a boost-coast thrust profile. Their characteristics are summarized in Figure 2. Reference missile AAM-2 was finally selected for replacing its rocket motor by a ducted rocket and simulate its performance with the performance code.

The performance gains offered by ducted rocket propulsion over the baseline solid rocket propulsion system have been evaluated for three mission scenarios (see Figure 3). The first mission scenario, a level flight, was selected as a basis for the design of the missile concepts, while the other two missions were used to verify the missile performance under high-dive and shoot-up flight conditions. The primary objective for performance improvements was set to a (conservative) value of 20% increase, relative to the range achieved with solid rocket propulsion. Though ramjet propulsion potentially allows for mass reduction at a similar flight envelope, this type of system improvement was not chosen as a requirement.



Original AAM-2 reference vehicle



IRR configured reference missile

Figure 2: Schematic drawing of the AAM-2 reference vehicle and the IRR reference missile.

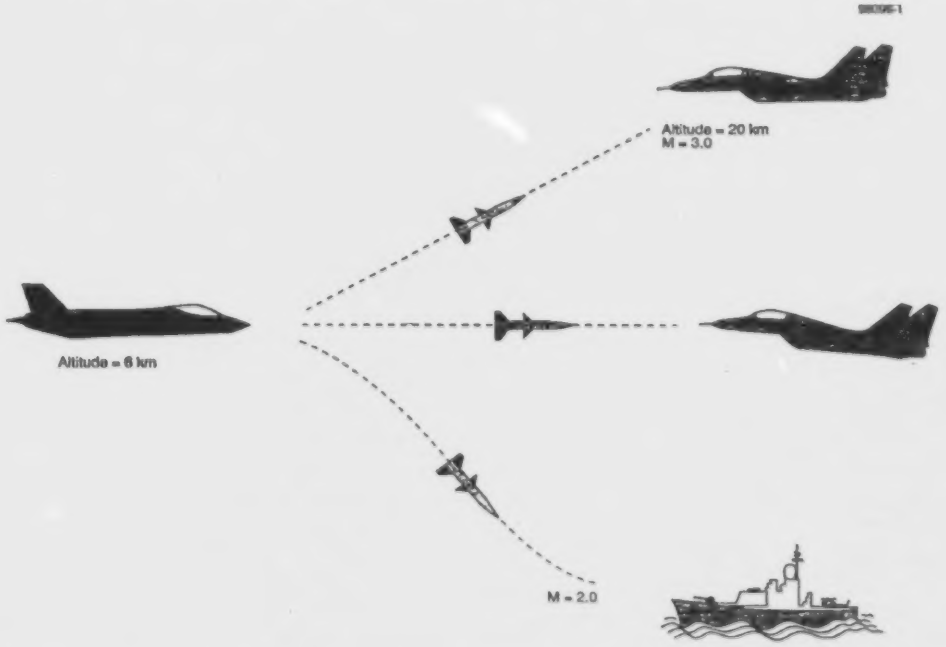


Figure 3: Reference missions considered as part of the IRRDP.

3 Missile design tool and performance prediction tool

Under the IRRDP, a software suite to support in-house research on ducted rocket propulsion was developed. Two complementary tools have been developed for the concept definition (the design tool [3] and [4]) and the performance prediction of a ducted rocket powered missile (the performance prediction tool [5], [6], [7] and [8]).

3.1 Design tool

For concept definition, the design tool generates a range optimized design of the ducted rocket powered missile based on specific geometry and performance requirements for a pre-determined mission profile. The engine design details the dimension, mass and on-design operating condition of the intake, gas generator, ramjet combustor and booster components and the net engine thrust output. The design tool incorporates models of sufficient accuracy for a relative evaluation of performance between the various configurations considered during the design optimization, such as shown below for a solid rocket motor (SRM) air-to-air missile concept and an integral rocket ramjet (IRR) air-to-air missile concept.

A detailed description of the physical models used, their functionality and limitations, and their integration to the design tool architecture is given in [3] and [4].

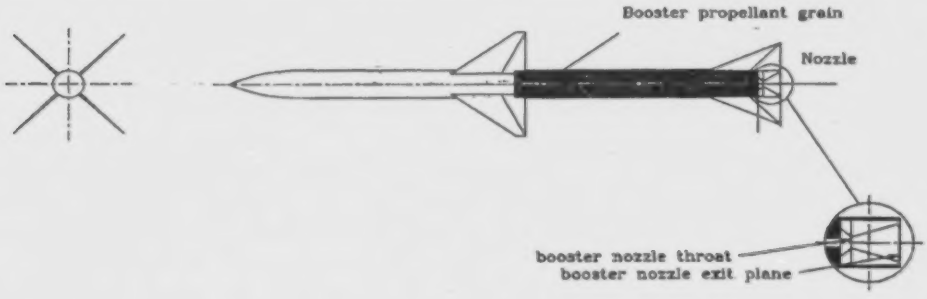


Figure 4: SRM concept.

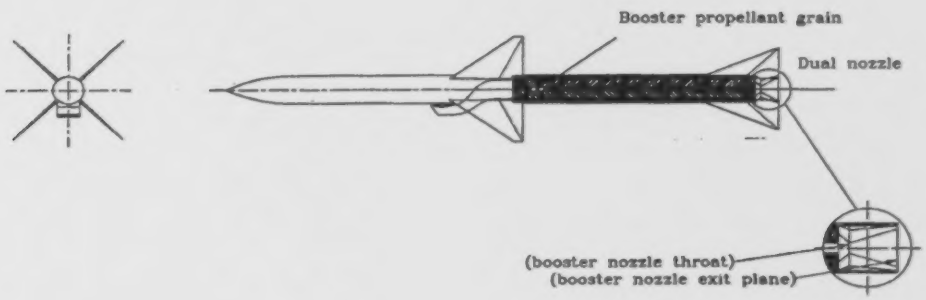


Figure 5: IRR concept.

3.2 Performance prediction tool

The performance prediction tool uses the output from the design tool as its own input, but allows system performance prediction with a higher level of accuracy than with the design tool. The model is able to predict the missile performance of a ducted rocket propelled missile, with the air intakes working in design as well as in off-design mode. The objective of the development of the performance prediction tool was to make it possible to:

- evaluate the performance of existing ducted rocket missile systems;
- evaluate the performance of future missile systems;
- fine-tune ducted rocket missile designs as created with the design model.

The model incorporates the three flight phases of a ducted rocket, being the boost phase, the ducted rocket propelled flight phase, and the non-propelled coast phase.

In the model, the ducted rocket is of the integral rocket ramjet type, which means the booster is integrated into the ducted rocket combustion chamber. The booster module is valid for a so-called nozzleless booster, and the air intakes are assumed to be of a rectangular shape. The model is able to calculate missile performance up to 32 km altitude, and treats the missile as a point mass moving in a two-dimensional plane. A detailed description of the possibilities, limitations and development of the performance prediction tool is reported in [5], [6], [7] and [8].

During a parametric study, the performance prediction tool proved to be a flexible and reliable tool to predict ducted rocket performance. Improvement of the missile trajectory calculations towards a six degree-of-freedom model is recommended to make end-game performance prediction possible. A typical result of the calculations with the performance prediction tool is shown in Figure 6.

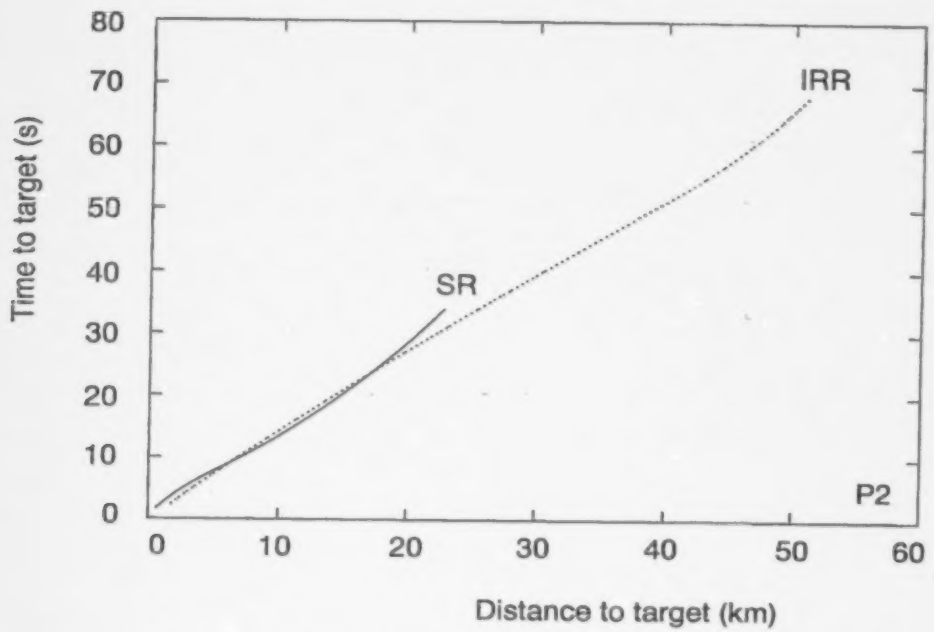


Figure 6: *Time to target for a solid rocket (SR) missile and a mass and geometrically equivalent integral rocket ramjet (IRR).*

4 Intake CFD and port covers

4.1 Intake CFD

The flow within a supersonic air intake is quite complex, being viscous, and can be laminar or turbulent. Shock waves are generated (oblique or normal) and shock-shock or shock-boundary layer interactions appear. If shock-boundary layer interactions are severe, the flow at the wall may separate and could choke the core flow if the separation bubble is reasonably large. This situation can also cause a high thermal load around the separated flow and may cause problems if local thermal loads exceed material limits. A systematic and comprehensive evaluation of the flow field in the engine diffuser inlet was therefore carried out for an accurate estimate of flight characteristics and performance [9].

Computational Fluid Dynamics (CFD) codes are now becoming efficient numerical tools to be used in the design of supersonic inlets. Their application can help cut down the time and cost of tests and help understand different phenomena from parametric studies. DREV used such codes to perform a study on three geometries of rectangular supersonic inlets. Two different codes were used; NPARC, which is a non-commercial code, and TASCFLOW, which is a commercial code. The codes were compared on their ability to calculate in a cost-efficient way the flow field around and in a two- or three-dimensional inlet, and to perform parametric studies. They were also compared for their ability to capture the shock waves in and around the inlet and to see the interactions between shock/shock and shock/boundary layer, and for their accuracy in calculating the back pressure needed to keep the normal shock wave at the throat as compared to that calculated from an analytical intake performance code supplied by TNO Prins Maurits Laboratory (TNO-PML).

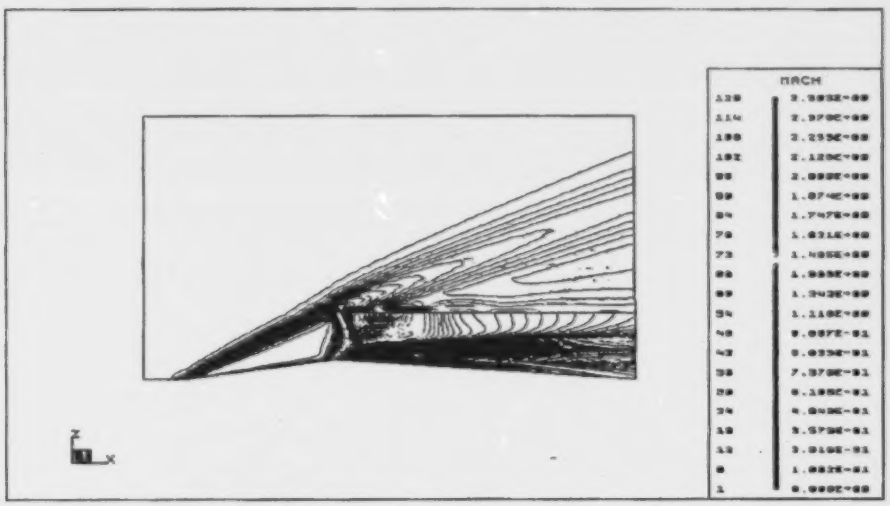


Figure 7: Example of a CFD calculation result for two-dimensional intake geometry (TNO-PML intake no. 2, viscous solution, Mach flow 2.5, p_{back} = 40 kPa).

The TNO-PML analytical code and a similar analytical DREV tool (programmed in the mathematical software package MATHCAD) were shown to be very good for calculating the back pressure needed for a critical operation. TASCFLOW was able to predict the general flow pattern and the location of the normal shock wave very well and was in good agreement with the analytical tools. The NPARC code was also good for flows that did not have recirculation zones at the outlet, but was poor when recirculation zones were present.

4.2 Port covers

Part of the work concerned the technology involved in port cover design. In order to become familiar with the different possibilities, a literature survey was carried out to determine the kinds of port covers that have been used [10].

The literature search was made using the DIALOG database. To this end, the NTIS, Aerospace Database (non-restricted version), SciSearch 1984-96, Ceramic Abstracts, Engineering Materials Abstracts and INSPEC databases were searched. Keywords were taken from the AGARD-CP-526 (Supp) Proceedings. The search showed that information on port cover development was scarce in relation to the number of articles available on ramjets. From the articles that were retrieved, several port cover concepts were discussed. These concepts were categorized as Monolithic, Frangible, Linear Shaped Charge, Hinged (Figure 8), Consumable, Flexible Membrane and Inlet. The descriptions for and the actuation mechanisms of the different concepts have been summarized in [10]. To aid in future trade-off studies, a list of selection criteria has been compiled from the articles themselves.

Figure 8: Hinged port cover.

5 Gas generator fuel development

A new solid propellant for a gas generator used in ducted rocket propulsion systems has been developed as part of the IRRDP. The propellant grain was based on Glycidyl Azide Polymer (GAP) formulations. Exploratory studies have been carried out [11] to assess the effect of variables such as the plasticizer type and concentration, the addition of a degradation catalyst or carbon black powder on the regression rate. Tests with each formulation have been performed using a strand burner. The effect of pressure on the burnrate was considered as well. It has been found that the addition of a plasticizer or carbon black powder to the gas generator propellant significantly increased the effect of pressure on the regression rate. However, the carbon powder decreased the regression rate. The overall azide concentration within the formulation was found to be a key factor controlling the regression rate. The tests also comprised mechanical properties evaluation in terms of viscoelastic dynamic moduli and glass transition temperature. A further investigation of the ballistic properties of the formulation was conducted in 02004 (ballistic) end-burner motors. These tests confirmed the trends observed in strand measurements (tests to determine the propellant burn rate), but the reproducibility between the two methods was found to be greatly affected by the experimental set-up and the preparation of samples. The final formulation retained for the program was found to have a regression rate slightly below the initial requirements of the program, but was demonstrated to be suitable for testing.

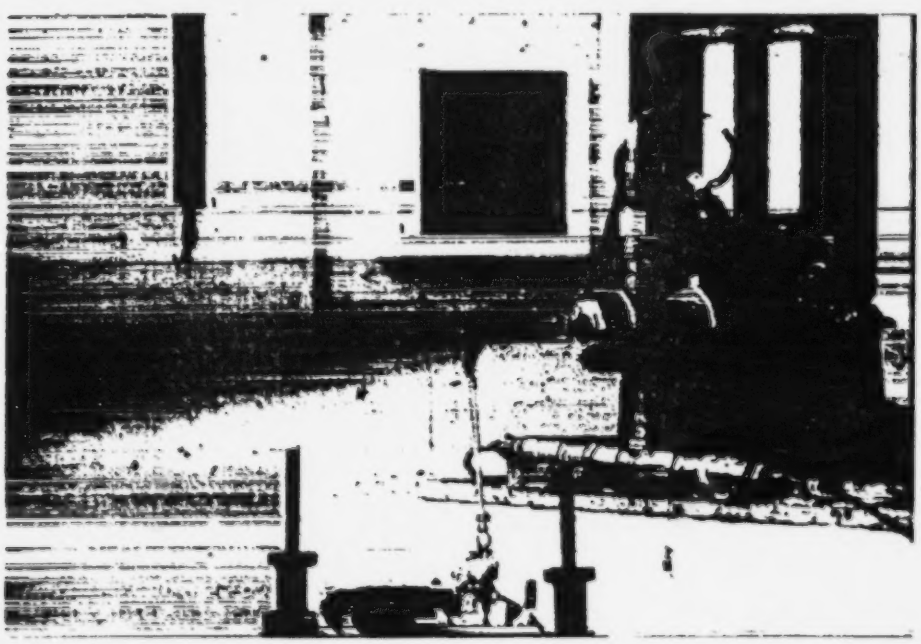


Figure 9: Firing of a 02004 ballistic motor filled with a GAP/Carbon black gas generator fuel.

6 Ramjet combustor

6.1 Water tunnel and CFD modeling comparison

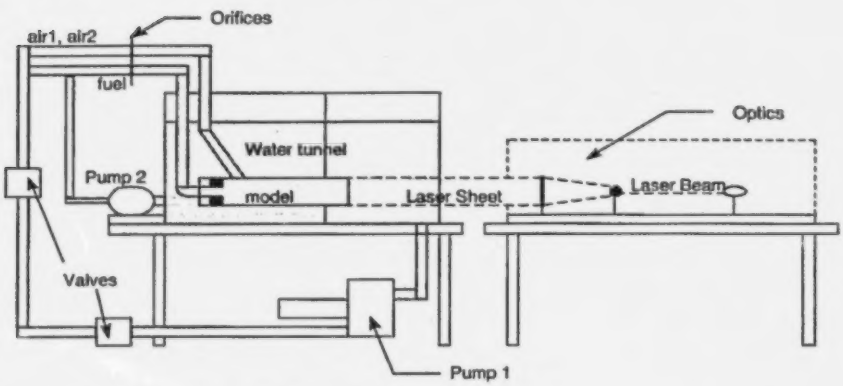


Figure 10: Schematic of the water tunnel.

The water tunnel is a 560 litre plexiglass tank in which a plexiglass model of a ducted rocket combustor is installed. It is an accepted method to visualize flow-fields within combustors, and has also been used specifically for ducted rocket combustors. It works by circulating water through the 'fuel' and 'air' inlets of the combustor to simulate the fuel and the air flowing through an actual combustor. To distinguish it from the airflow, the fuel flow is seeded with very fine cavitation bubbles. A laser sheet illuminates a plane of interest inside the combustor so that the 'fuel' and 'air' mixing can be easily seen and recorded with a video camera.

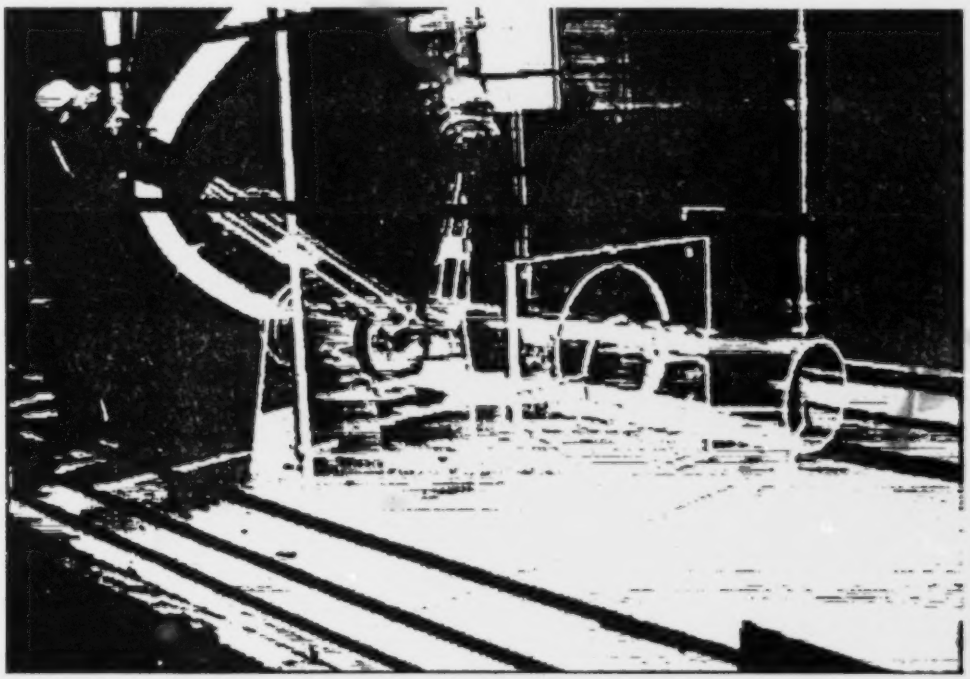


Figure 11: Plexiglass model of combustor

The main advantage of a water tunnel for flow visualization, apart from the fact that the model combustor is transparent, is that the Reynolds numbers achieved for a water speed in the main part of the combustor of 1 m/s are representative of the flow in a typical ducted rocket combustor, and the flow patterns are easily seen in real time with the human eye. Several combustor models were built and tested at various flow conditions (fuel/air momentum ratios) to encompass all the practical possibilities for testing at TNO's direct-connect combustion test facility. Figure 12 shows an image from test configuration 7 where the light-coloured bubbles representing the fuel can be seen. The area with more bubbles represents the higher concentrations of fuel. Qualitatively, these images compare well with the results of the computational fluid dynamics (CFD) modeling of the same configurations, particularly when the lower intensity of the laser sheet at the top and bottom and the effects of forward- and back-scattering are considered. However, a very important difference is that the water tunnel is able to show time-dependent phenomena which occur in an actual ducted rocket combustor, while the CFD modeling assumes quasi-steady flow.

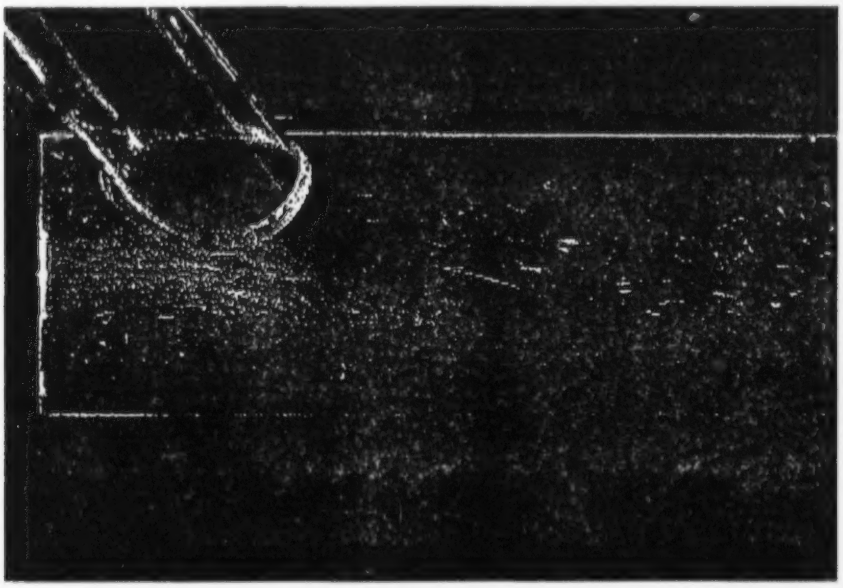


Figure 12: Water tunnel image, mixing on the centreline plane, configuration 7.

The CFD modeling of the water tunnel experiments was carried out using a code known as FLUENT (Version 4.4). This is a three-dimensional, finite volume code which uses discretized Navier-Stokes equations to solve incompressible or mildly compressible non-reacting or reacting flow problems, both laminar and turbulent. It requires a structured grid of the geometry of interest. The basic geometry is shown in Figure 13. The first task with this code was to model all the ducted rocket combustor configurations tested in the water tunnel as turbulent, non-reacting flow, and compare these with the water tunnel images. Figure 14 shows the results of the modeling for configuration 7 for fuel concentration on the centreline plane and compares this with the water tunnel image for the same configuration. In contrast to Figure 12, however, the water tunnel image has been corrected so that a direct quantitative comparison between the two methods can be made. The scale of fuel mass fraction is also shown. The red shows the highest fuel concentration, followed by green and light blue, while dark blue indicates an absence of fuel. If the upper left corner of the water tunnel image is neglected because of reflections from where the air inlet meets the combustor, very good agreement is obtained, considering time-dependency as well.

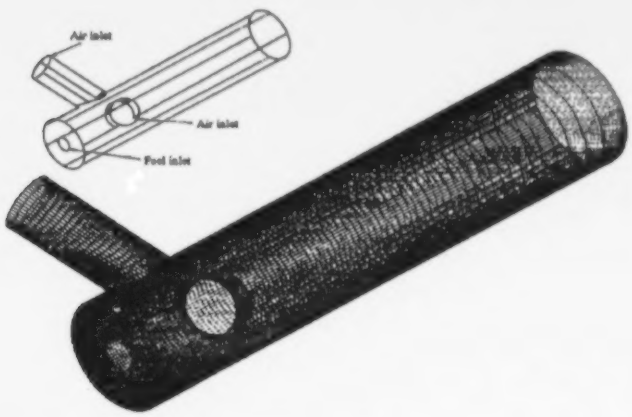


Figure 13: Basic geometry.

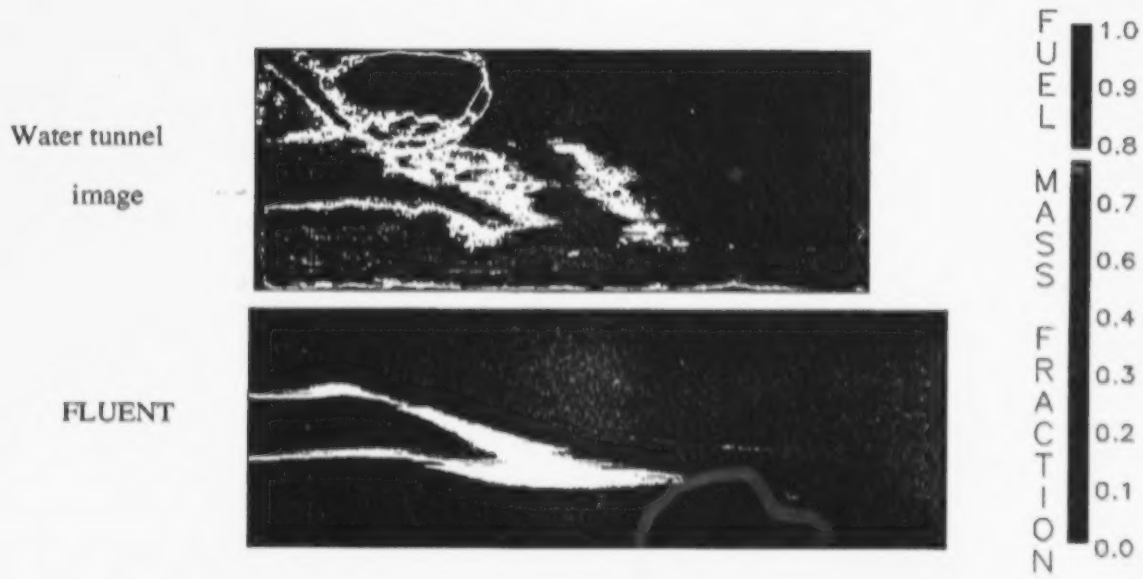


Figure 14: Fuel mass fraction, configuration 7, air/fuel momentum ratio = 3.04.

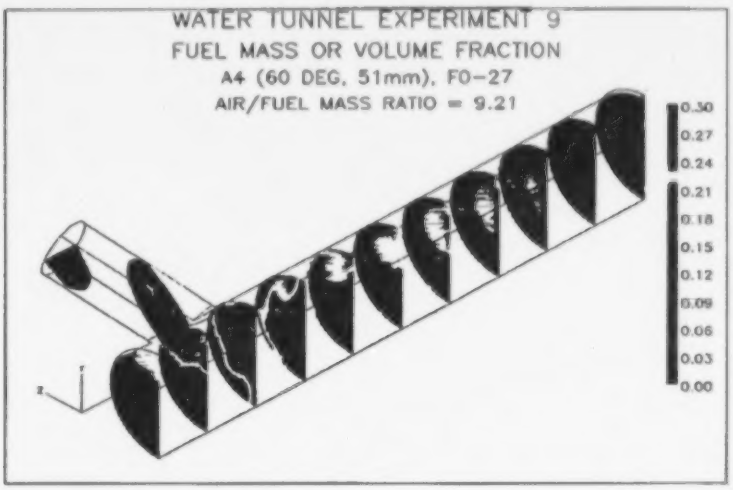


Figure 15: Fuel mass fraction, configuration 9.

FLUENT was then used to model other combustor geometries of interest. This is because despite its limitations due to the accuracy of certain assumptions, such as quasi-steady flow, and submodels, CFD modeling was able to evaluate a wide range of geometries and test conditions more quickly than with the water tunnel and combustion tests. It can also provide more comprehensive data which would be difficult or impossible to measure experimentally. However, the limitations mentioned above mean that at least some experiments are still essential for validation purposes. Figure 15 shows a contour plot of fuel mass fraction at various cross-sections for a fuel injector with a fence that has not yet been experimentally tested. The results show that while some of the fuel mixes rapidly in the head end, which is desirable, the mixing is not so good downstream from the air injectors. Once the water tunnel experiments were all modelled, another hypothesis was checked. This hypothesis was that apart from geometry and Reynold's number, the ratio of the air and fuel momentums must be identical to reproduce the same flowfield and fuel/air mixing, in the absence of combustion, in the water tunnel and the actual combustor. This was modelled by replacing the water used to simulate the fuel and the air by data for actual fuel and air at the same momentum ratios. Except for some minor differences where velocities were low, the flowfields from the CFD analysis were the same. This supports the use of the water tunnel to simulate the non-reacting flowfield in an actual ducted rocket combustor, as long as the geometry and the momentum ratio is kept the same and the Reynold's numbers are sufficiently high. References: [12] and [13].

6.2 Scaling

While CFD modeling and water tunnel experiments can provide much useful data on the mixing processes inside a ducted rocket combustor [12] and [13], experi-

ments on an actual, operating combustor are essential to provide reacting flow data and performance measurements. Such tests were done at TNO-PML's direct-connect combustion test facility [14]. In the case of the ducted rocket combustor, existing hardware based on 100 mm diameter was already available at TNO-PML (Figure 16), so in order to save costs this was used in the IRRDP test program. In general, however, propulsion systems for missiles of interest are of a larger diameter. The test results on the smaller 'sub-scale models' were therefore scaled up to the larger, 'full-size' combustor.

The scaling of test results has been done extensively in the field of fluid mechanics for non-reacting flows [15]. Through the use of dimensionless parameters such as the Reynold's number, experiments involving similar geometries, but having different sizes and fluid properties, can be directly compared with each other if the values of the dimensionless parameters are maintained. The use of dimensionless parameters and scaling can also be applied to reacting flows once chemical kinetics, species transport, and thermal processes are also considered. It has been applied in the past to gas turbine, liquid fuel ramjet, solid fuel ramjet, and scramjet combustors.

To ensure that the most important dimensionless parameters are maintained in the sub-scale and full-size combustors, PD-scaling has often been applied. This type of scaling requires that geometric similarity be maintained, and that the product of the static pressure in the combustor and the length scale (diameter) be held constant between the model and full-size (prototype) combustors:

$$p_m d_m = p_p d_p$$

where p is the static pressure, d is the diameter, and the subscripts m and p are for the model and the full-size combustor (prototype), respectively. Inlet velocities, fuel type, air/fuel ratios, and inlet temperatures must also be kept constant. Assuming that the global reaction rate is second order and that heat losses through the combustor wall are not significant, then PD-scaling should allow test results from the sub-scale ducted rocket combustors to be extrapolated to the larger propulsion systems of interest.

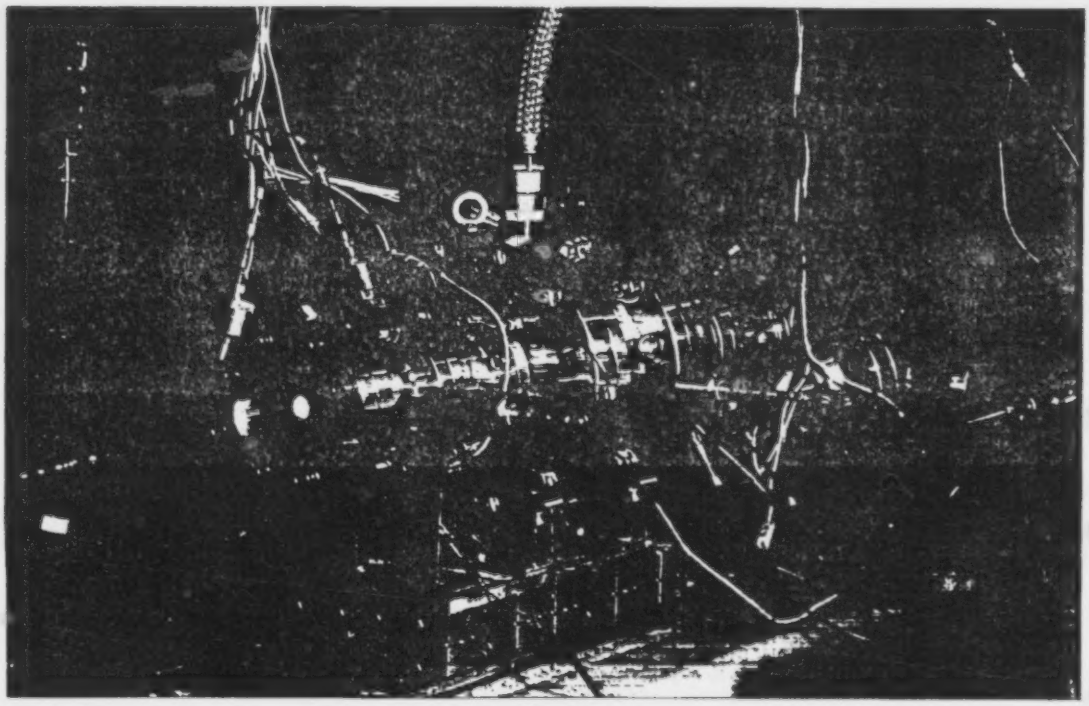


Figure 16: Sub-scale combustor test hardware (931012-28).

If PD-scaling is applied as described above, assuming that the global chemical reaction rate for the combustion is second order and that heat losses through the combustor wall are not too important or scale properly, then most of the derived dimensionless parameters will be similar in the model and the prototype combustors. The remaining dimensionless parameters can be neglected for ducted rocket combustors, since turbulent mixing should overwhelm molecular mixing, buoyancy effects should be small, and convective heat transfer should be much more important than radiation. If the above are all true, then PD-scaling should be applicable to our ducted rocket combustor tests.

6.3 Combustion tests

Within the IRRDP, combustion experiments were performed to obtain data on the combustion characteristics inside a ducted rocket combustion chamber [14]. Also the experiments were done to select the optimal fuel/air injector combination for the ducted rocket motor. The combustion experiments took place at the TNO-PML connected-pipe ramjet test facility. For the tests at TNO-PML, an existing ducted rocket test motor was used with some adaptations. The most important adaptations to the test facility are summarized below:

- addition of an ethylene gas supply system to the test facility. With this, the products from the decomposition of the solid fuel grain in the gas generator are simulated by the combustion of a very rich ethylene/air mixture. This greatly reduced the number of gas generator grains needed and reduced test costs;
- a new stagnation vessel (plenum chamber) was built to ensure a proper airflow through the ducted rocket air intakes;
- new fuel and air intakes were built to obtain realistic motor configurations.

The fuel used during the ramjet test at the TNO-PML connected-pipe test facility as shown in the following figure was the ethylene/air mixture to simulate the GAP solid fuel as manufactured at DREV. The temperature of the air supplied to the test motor was about 600 K to simulate a Mach 2.5 flight.

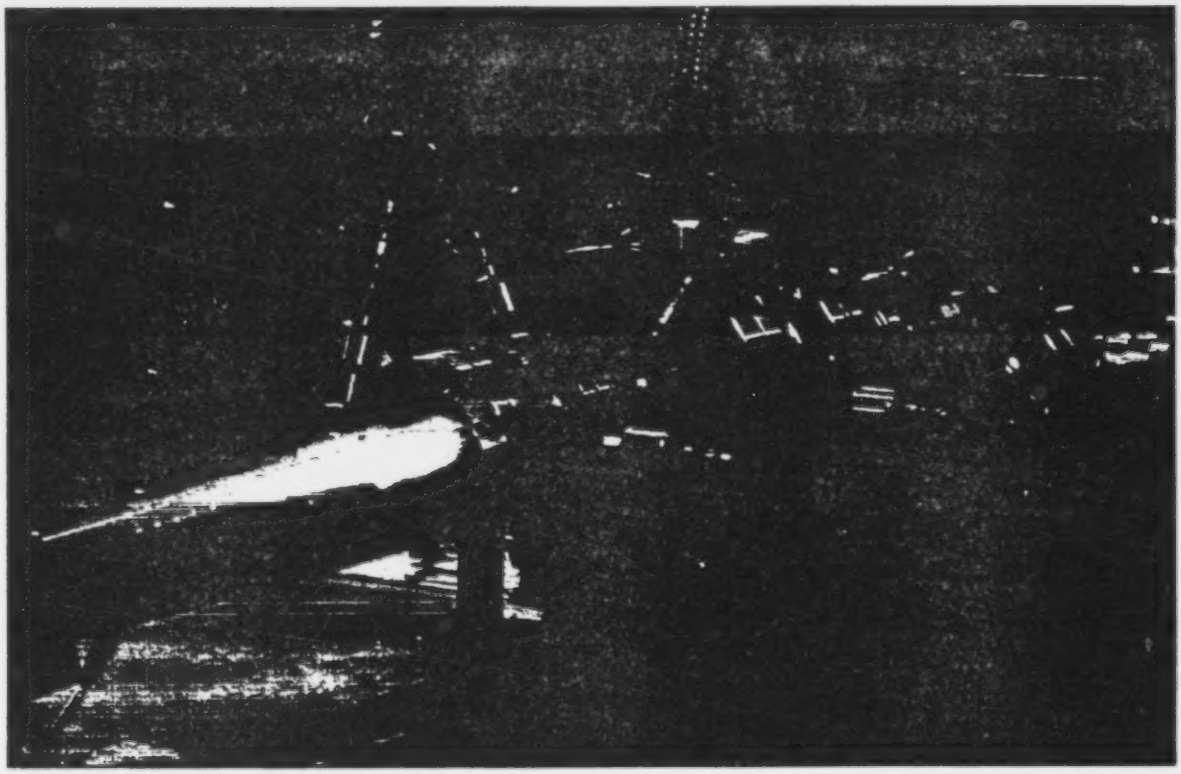


Figure 17: Test firing of the ducted rocket motor.

During the experiments four air injectors and four fuel injectors were extensively tested. These tests resulted in the following optimum air and fuel injector combinations:

- air injector: injection angle 60° and with two 38.7mm diameter tubes;
- fuel injector: a single 18mm diameter fuel port centrally located.

The direct-connect combustor test results were compared with the results obtained with the water tunnel experiments and CFD calculations. Agreement between the three methods was good. Combustor performance was found to be highly dependent on the test conditions (e.q. flight conditions) such as chamber pressure, air mass flow rate, fuel mass flow rate, air temperature, etc.. Since combustor performance is an extremely important input for missile performance simulation with the performance computer model, the data from the ducted rocket combustor experiments are of great value.

7 Nozzleless booster

7.1 Nozzleless booster model and CFD

Since a ramjet can only make sufficient thrust at typical speeds above Mach 1.2 to 1.4, a rocket must be used to boost it from launch velocity. A very attractive option to do this is an integral nozzleless booster. Work during the IRRDP on such boosters included the development of a nozzleless booster module for the performance prediction tool [16]. It calculates the performance requirements needed for a missile to achieve a specific burnout velocity before ramjet combustor ignition. To validate the model, some preliminary studies were performed using computational fluid dynamics to depict the flow contours from such a booster.

The nozzleless booster module, which is based on both theoretical and experimental analysis done in open literature, calculates the interior ballistic requirements for a predetermined sized booster and initial launch conditions. The analysis is based on a quasi-steady-state one-dimensional flow of a perfect gas in a constant area port, assuming a uniform burning rate along the grain of the propellant. Given the initial bore diameter or chamber pressure, the thrust, mass flow, exhaust speed, speed of vehicle, and specific impulse are calculated with respect to time of burn. It has also been shown that the length-to-bore diameter ratio of the booster does effectively affect the thrust and consequently the speed of the vehicle. It was found that increasing this ratio increases the mass flow, thrust and speed and decreases the burn time for a specific propellant operating at a fixed chamber pressure. Figure 18 shows the effect of increasing the L/D ratio on the thrust of the booster. However, in order to assess the most efficient configuration, such a study must consider the overall vehicle requirement. This study will be continued within the 6DOF work of the follow-on program.

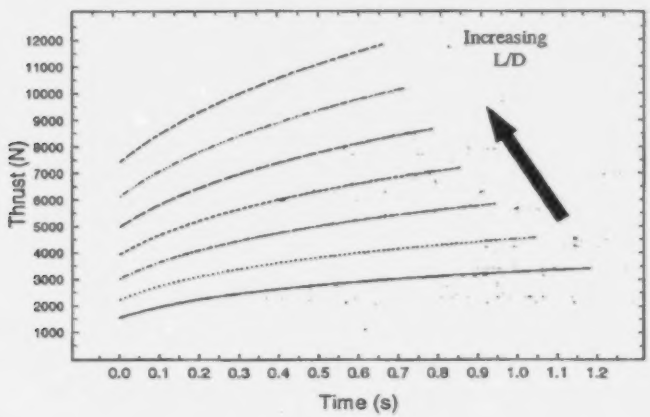


Figure 18: L/d effect on thrust as calculated by the numerical model in the performance code.

The CFD performed on the nozzleless booster attempted to validate the assumptions and theory presented within the module. The numerical analysis included a simple 3D laminar incompressible flow model using air as the flow species. The computer code used for solving the flow was TASCFLOW. The model inputs were the chamber pressure and the bore diameter. The code generated the flowfield through the bore diameter and at the exhaust. It was shown that both the performance code module and the CFD calculated similar exhaust speeds. Similar results were obtained when boundary conditions included mass flow rather than pressure inputs. This latter condition showed a more realistic flow field through the bore diameter. Typical results are shown in Figure 19. The bottom part of this figure represents a more realistic flow due to the increasing velocity rather than constant velocity through the bore.

Figure 19: Velocity contours from 3D CFD simulations.
Top : simulation for a pressure inlet condition.
Bottom : simulation for mass flow inlet conditions.

7.2 Propellant development for a nozzleless booster

The experimental investigation for nozzleless boosters included the development of a high burn rate propellant. One of the significant challenges is to produce a propellant with a high enough burn rate to cause the exhaust gases to choke at the exit of the booster for a given length-to-bore ratio. As a starting point, an HTPB/AP propellant formulation from a previous project on high burn rate propellant was selected. To raise the burn rate even further, work focused on using different burn rate catalysts and very fine grinds of ammonium perchlorate (AP) oxidizer. However, problems with very high viscosity during mix cycle and high end-of-mix viscosities (>10 kP) were experienced due to the use of this fine AP. To correct this, the options studied were the use of reduced percentage of fine AP, a lower solid content, a modified mix cycle and an increased binder plasticizer content. In addition, different additives were studied to increase the burn rate and lower pressure exponent at low pressures.

Promising formulations were chosen for scale-up from small to larger batches. Strand burn rate results are illustrated in Figure 20. Burning rates of 33-35 mm/s at 6.89 MPa are obtained with a rather high pressure exponent in the 0.76-78 range between 3.45 and 13.79 MPa pressure range (500-2000 psia).

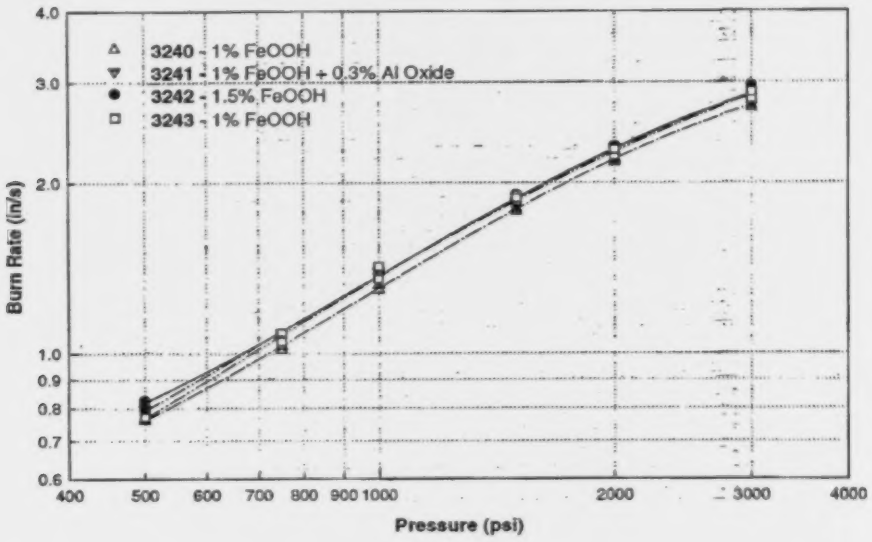
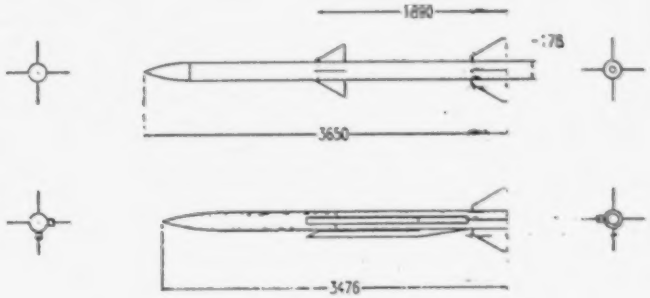


Figure 20: Strand burn rates of sub-scale batches.

An additional sub-scale 3 kg batch has now been processed and several burn rate motors cast to assess the ballistic behaviour of the propellant formulation in actual motors. If the high exponent is judged tolerable from the results, the casting of actual nozzleless booster motors will take place without further modifications to the formulation. There are plans to test these in the future.

8 Final design

The IRRDP study was concluded with the collection of all results obtained and combining them into an updated reference missile design (included in [8]). With the performance model, the performance of the updated missile design was determined. With respect to the solid rocket missile system, an 85% improvement in range was obtained, even though the booster and gas generator propellants did not completely meet their desired specifications. The 85% increase in range is well above the desired requirement of a 20% increase in range. If the booster and gas generator propellants had been able to meet the requirements as set at the beginning of the IRRDP, then an improvement in range of 110% would have been obtained. Figure 21 shows the designs of the solid rocket reference missile and its updated IRR equivalent. The geometry of the ducted rocket combustion chamber is shown in Figure 22.



Reference (Rocket)

Integral Rocket Ramjet

Figure 21: Design of the reference solid rocket missile and its upgraded IRR equivalent.

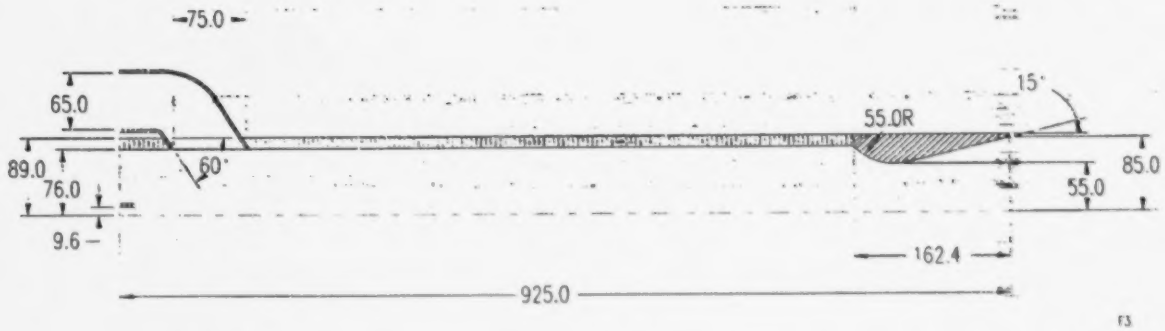


Figure 22: Final design of the ducted rocket combustion chamber.

For the updated IRR design, the following three missions were flown with the performance tool as specified in the requirements. The missions were:

- a level flight at 6 km altitude (reference mission). After boost acceleration until the flight Mach number is 2.5, after this continuing with a constant velocity flight;
- a descending flight from 6 km altitude to sea level to reach a flight Mach number of 2.0 (1);
- a climbing flight from 6 km altitude to 20 km altitude to reach a flight Mach number of 3.0 (2).

The updated IRR missile was able to fly all three missions as can be seen in Figure 23.

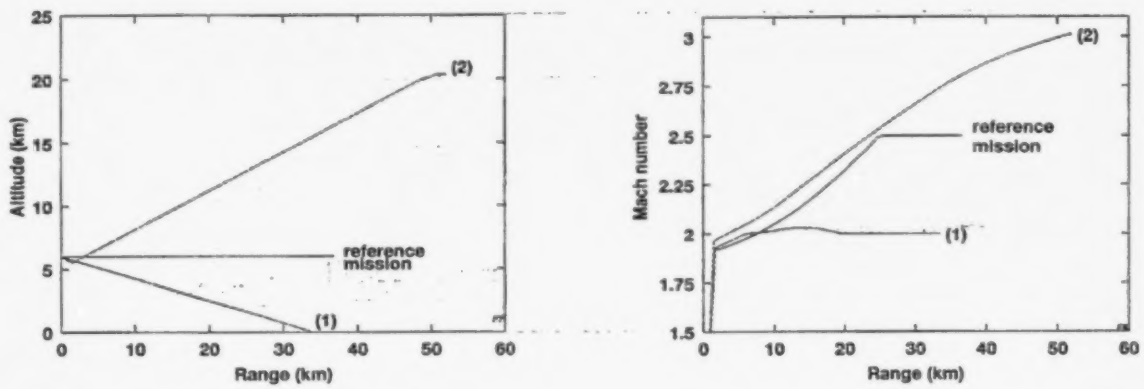


Figure 23: With the ramje- propelled missiles the requirements are met for all three missions.

9 Concluding remarks

The main technical accomplishments of the IRRDP were:

- definition of the requirements for a medium-range air-launched missile;
- computer software tools for the design and performance prediction of ducted rocket powered missiles;
- analytical and CFD modeling of air inlets;
- a survey of port cover technologies;
- development, fabrication, and testing of a gas generator fuel based on glycidyl azide polymer (GAP);
- determining the effects of geometry and test conditions on combustor performance, through computational fluid dynamics (CFD) modeling, water tunnel flow visualization, and sub-scale combustor studies;
- analytical and CFD modeling of nozzleless boosters.

The success of the program has led to the consensus in both countries that follow-on work is highly desirable. Approval and funding for a three year follow-on program has since been granted. The follow-on program will focus on improving the performance prediction code as developed under the IRRDP and will include:

- improved flight trajectory modeling (6DOF);
- improved missile aerodynamics;
- add simple guidance and control routines;
- include several types of ramjet;
- include several types of booster.

This follow-on program will continue until December 2000.

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11 Authentication

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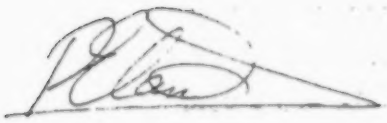

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15. ABSTRACT (MAXIMUM 200 WORDS (1044 BYTE))

The Defence Research Establishment Valcartier (DREV) in Canada and the TNO Prins Maurits Laboratory (TNO-PML) in the Netherlands have undertaken a collaborative activity known as the Integral Rocket Ramjet Demonstration Program (IRRDP) to investigate the key technologies of airbreathing missile propulsion through the design, fabrication, and testing of a solid fuel ducted rocket engine with an integral booster for a medium-range air-launched missile. Falling under the CA/NL Memorandum of Understanding on Co-operation in Defence Science and Technology, the resources expended on each side were approximately equal, being equivalent to one million NLG or $750K Canadian. The program's goal was to develop a technology base for the Canadian and Dutch Armed Forces in order to meet the following objectives:

- to improve the operational capabilities of the Armed Forces, in terms of threat assessment, foreign weapons evaluation and potential applications;
- to be able to assist the Armed Forces in decisions concerning future purchases of weapon systems equipped with airbreathing propulsion, and;
- to be able, if an international industrial effort is initiated, to participate in future multi-national (airbreathing) propulsion projects for missile development programs on a basis of (industrial) expertise and capability and demonstrated international co-operation.

The report describes each of the main accomplishments of the IRRDP in more detail, including descriptions of how the work was carried out and examples of the results.

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Intgral rocket ramjets	Missiles	Ramjet propulsion	

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TITLE:(U) INTEGRAL ROCKET RAMJET DEMONSTRATION PROGRAMME (IRRDP). EXECUTIVE SUMMARY REPORT

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ABSTRACT: (U) The Defence Research Establishment Valcartier (DREV) in Canada and the TNO Prins Maurits Laboratory (TNO-PML) in the Netherlands have undertaken a colaborative activity known as the Integral Rocket Ramjet Demonstration Program (IRRDP) to investigate the key technologies of airbreathing missile proplusion through the design, fabrication, and testing of a solid fuel ducted rocket engine with an integral booster for a medium-range air-launched missile. Falling under the CA/NL Memorandum of Understanding on Co-operation in Defence Science and Technology, the resources expended on each side were approximately equal, being equivalent to one million NLG or $750K Canadian. The program's goal was to develop a technology base for the Canadian and Dutch Armed Forces in order to meet the following objectives: - to improve the operational capabilities of the Armed Forces, in terms of threat assessment, foreign weapons evaluation and potential applications; - to be able to assist the Armed Forces in decisions concerning future purchases of weapon systems equipped with airbreathing propulsion; - to be able, if an international industrial effort is initiated, to participate in future multi-national (airbreathing) propulsion projects for missile development programs on a basis of (industrial) expertise and capability and demonstrated international co-operation. TRUNCATED

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